

Sub-Payload Landing Technology for the L4 Enceladus Lander

Concept trade and first design iteration of a post-landing deployment system (PLDS)

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Contents

1	Introduction	2
2	Motivation and requirements	2
3	Assumptions and baseline parameters	2
4	Concept discussion	2
4.1	Deployment during descent (DDD)	2
4.2	Deployment after landing (PLD)	3
4.3	Further concepts and variations	3
5	Analysis	3
5.1	Deployment during descent	3
5.2	Deployment after landing	5
6	Discussion and selection of the design baseline	6
7	Design of the post-landing deployment system	7
7.1	Means of propulsion	7
7.2	Mechanical and functional concept	7
7.3	Springs	9
7.4	Guidance	10
7.5	Hold-down and release mechanism	10
7.6	Pedestal	10
7.7	Mass budget	11

1 Introduction

This technical note gives an overview of the challenges and options for landing small sub-payloads at a distance from the primary Enceladus lander of the ESA L4 mission. It derives a design baseline from a concept trade and first-order analysis, and documents a first design iteration of the selected deployment system.

2 Motivation and requirements

The need for landing sub-payloads stems primarily from two instrument needs presented at the DGP 2026 L4 workshop:

- creating one or more extended baselines of multiple receiver nodes to enhance ground-penetrating radar (GPR) sounding (cf. the CORSE surface GPR network entry);
- inserting instrumentation into environments that are not suitable as primary landing zones (e.g. tiger stripes, steep terrain).

This leads to the need to land payloads of typically below 1 kg at distances of several hundred metres to potentially kilometres from the main lander. The following discussion draws mainly on the needs of a potential GPR payload, which should however not limit the conclusions drawn.

3 Assumptions and baseline parameters

Results are presented parametrically where practical. The baseline parameter set is given in Table 1.

Table 1: Baseline assumptions.

Parameter	Value	Remark
GPR receiver node	0.5U, 370 g	Without landing hardware
Moving mass for sizing	0.5 kg	Includes margin
Lander payload budget	10 kg total	Mass is a primary driver
Lander total mass	600 kg	
Surface gravity g	0.113 m/s ²	
Descent profile	Gravity turn, constant thrust acceleration	Reality will differ
Geometry	Local flat surface	Curvature and Coriolis effects below 1% at 500 m
Environment temperature	≈ 75 K	
Separation from lander	≥ 250 m, target 500 m	
Number of nodes	1 or more, target 4	

4 Concept discussion

Three top-level sub-landing concepts can be distinguished:

1. landing independently after separation from the orbiter;
2. deployment from the lander during the descent phase (DDD);
3. deployment from the lander after touchdown (post-landing deployment, PLD).

The first concept is discarded because it requires dedicated descent propulsion, effectively making the sub-payload a lander in its own right. Viable solutions for this approach are conceivable, but it is not considered further here because of its inherent complexity.

4.1 Deployment during descent (DDD)

Releasing the sub-payload during the descent of the primary lander is attractive for several reasons:

- no additional source of propulsive energy needs to be carried;

- large throw distances can be achieved;
- the throw distance can be adjusted, even late, through the choice of separation time;
- deployment does not depend on a successful touchdown of the primary lander. This advantage is weaker than it appears, however: if the nodes rely on the lander as GPR transmitter, data relay or power source, a node delivered without a functioning lander returns little or no science.

These advantages come with a number of disadvantages:

- resulting impact velocities are likely high (see Section 5.1);
- unless additional equipment is used, sub-payloads land only along the principal approach axis of the main lander, which makes the concept less attractive for achieving non-degenerate baselines;
- the risk of the deployment interacting with the main lander must be mitigated (separation shock, collision risk, interaction with the descent engine plume if firing).
- impact occurs at a shallow flight-path angle (about 6° to 11° above the horizon for the cases in Table 2), which promotes skipping and tumbling on the surface (this might not be a primary concern though).

4.2 Deployment after landing (PLD)

Post-landing deployment is almost the opposite of DDD. Its advantages are:

- the descent and landing of the primary lander are fully decoupled from the sub-payloads, which reduces mission-planning complexity and risk; the sub-payload system remains self-contained in function and performance;
- throw distance and impact velocity can be tailored with relatively high precision;
- impact velocity is low, at least for moderate separation from the lander;
- the lander can potentially observe the trajectory and impact point.

Its disadvantages are:

- propulsive energy must be carried and stored until deployment;
- deployment is strictly local to the lander and depends on the landing attitude;
- the throw distance is limited (likely several hundred metres, see Section 5.2);
- a larger mass allocation is needed because of the more substantial deployment system.

4.3 Further concepts and variations

Some drawbacks of the DDD concept can be addressed with a twin sub-payload that is first separated from the main lander and then pushed apart, e.g. by a second separation spring, adding a lateral vector to the landing pattern. Such lateral deployment can also avoid falling through the descent engine plume, but it adds complexity and likely removes the mass advantage of DDD.

5 Analysis

5.1 Deployment during descent

The fundamental question is: *when does a payload have to be released from the main lander to achieve a given separation from the main landing point, and what are its impact conditions?* Atmospheric effects are absent on Enceladus, and a flat-world approximation is sufficient for a first evaluation, which leads to simple closed-form solutions.

5.1.1 Ballistic segment after release

Consider release at altitude h_0 above the local surface with horizontal velocity v_h and descent rate v_d (positive downward). The time to impact t_i follows from $h_0 = v_d t_i + \frac{1}{2} g t_i^2$:

$$t_i = \frac{-v_d + \sqrt{v_d^2 + 2gh_0}}{g} \quad (1)$$

The impact velocity components are

$$v_{d,i} = v_d + gt_i = \sqrt{v_d^2 + 2gh_0}, \quad v_{h,i} = v_h \quad (2)$$

and the horizontal distance covered after release is

$$x_i = v_h t_i. \quad (3)$$

The quantity of interest is the separation from the main lander's touchdown point, $\Delta = x_i - x_L$, where x_L is the downrange distance the lander still covers between the release point and touchdown.

5.1.2 Gravity-turn descent model

These relations are combined with a trajectory model of the main lander's terminal powered descent, assumed to be a constant-thrust-acceleration gravity turn (a short deorbit burn earlier brings the lander out of its initial Enceladus orbit after separation from the orbiter). Since any practical Enceladus lander achieves a very high thrust-to-weight ratio with respect to local gravity, an extended hover phase for final landing-site selection and hazard avoidance is likely feasible. Because this phase is irregular, it is assumed that the sub-payload separates before it, while a guaranteed forward velocity still exists. Whether this assumption is reasonable needs further investigation.

With φ the angle of the velocity vector from nadir (horizontal flight is $\varphi = 90^\circ$), $a = ng$ the thrust acceleration and n the thrust-to-weight ratio, the gravity turn obeys

$$\frac{dv}{dt} = -a + g \cos \varphi, \quad v \frac{d\varphi}{dt} = -g \sin \varphi. \quad (4)$$

Dividing the two equations gives $d(\ln v)/d\varphi = n/\sin \varphi - \cot \varphi$,

Integrating from a reference state (v_1, φ_1) gives Equation 5. The horizontal velocity takes a particularly simple form, which makes it easy to find the release point for a given v_h . Position and time follow by quadrature, $dt = -v d\varphi / (g \sin \varphi)$, $dx = v \sin \varphi dt$, $dz = v \cos \varphi dt$. The model assumes constant thrust *acceleration* (thrust throttled with decreasing mass); for constant thrust, n grows slowly during the burn.

$$v(\varphi) = v_1 \frac{\sin \varphi_1}{\sin \varphi} \left[\frac{\tan(\varphi/2)}{\tan(\varphi_1/2)} \right]^n, \quad v_h(\varphi) = v \sin \varphi = v_{h,1} \left[\frac{\tan(\varphi/2)}{\tan(\varphi_1/2)} \right]^n \quad (5)$$

5.1.3 Results

A numerical evaluation gives the throw range for release below a given horizontal velocity (Figure 1). In this case a gravity-turn descent with $n = 10$ is modelled and hypothetical sub-landers are released below 50 m/s horizontal velocity.

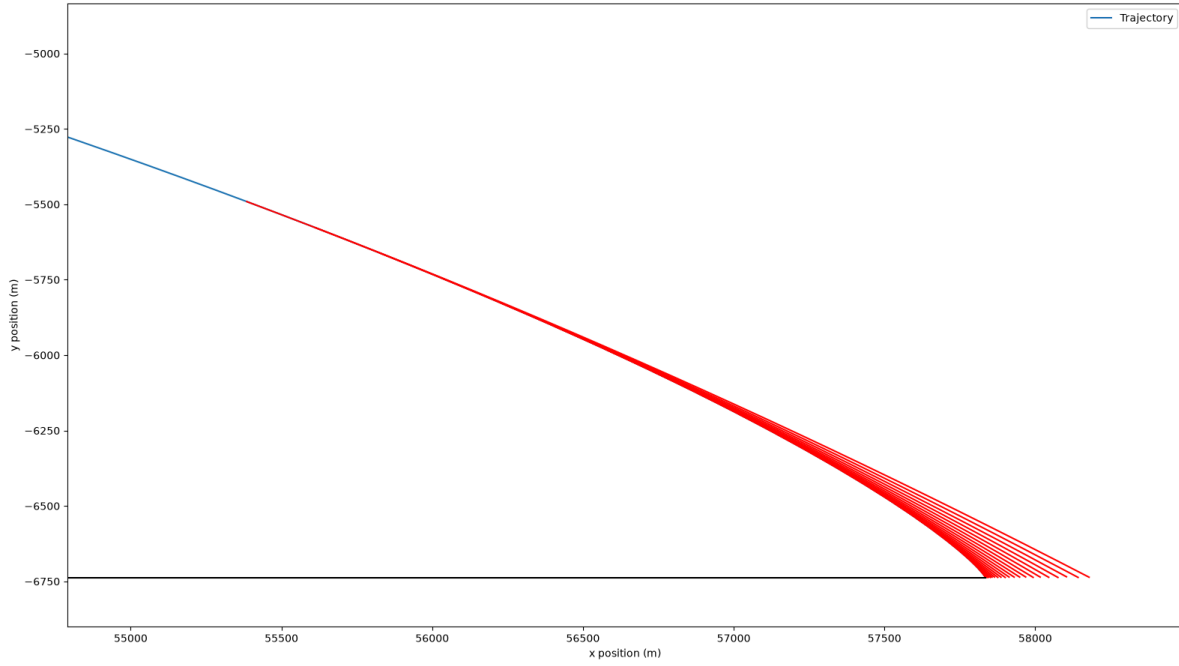


Figure 1: Gravity-turn descent ($n = 10$) of the main lander (blue) and ballistic trajectories of sub-payloads released at horizontal velocities below 50 m/s (red). Axes in metres; vertical axis is not to scale with the horizontal axis.

In the case shown in Figure 1, the most distant sub-lander impacts about 340 m from the main lander touchdown point. This satisfies the 250 m minimum separation but not the 500 m target, and the impact occurs at more than 50 m/s, which likely adds additional engineering challenges.

To generalise this result, Equation 1 to Equation 5 were evaluated for $n = 10$ and three gravity-turn entry speeds; the results are given in Table 2, with the separation Δ measured from the lander touchdown point. The impact speed is essentially equal to the horizontal speed at release, because the descent rate at release is small. A separation of 250 m requires release at about 50 m/s, and the 500 m target requires release at 66 to 80 m/s, with impact at the same speed regardless of the entry conditions. The impact energy is then $(50/7.5)^2 \approx 44$ to $(80/7.5)^2 \approx 114$ times that of the post-landing case (Section 5.2).

Table 2: Sub-payload release during an $n = 10$ gravity turn with $g = 0.1135 \text{ m/s}^2$. Ranges cover gravity-turn entry speeds (horizontal flight) of 100, 150 and 200 m/s. For a given release speed, larger entry speeds give higher release altitudes but smaller separation; the first column gives the release conditions that achieve the 500 m target.

v_h at release (m/s)	66–80	50	40	30	20	10	5
Release altitude (m)	178–408	134–214	102–154	69–99	39–52	13–17	4–5
Separation Δ (m)	500	151–228	88–125	44–60	17–22	4	1
Impact speed (m/s)	67–81	50–51	40–41	30–31	20–21	10–11	5–6
Impact angle above horizon	6°–9°	7°–11°	9°–12°	10°–14°	12°–16°	16°–20°	20°–24°

Further parameter studies are possible, but under a gravity-turn descent it is unlikely that a deployment configuration combines adequate throw range with reasonably low impact velocity. Descent strategies with a hover or translation phase at (near) constant altitude and slow to moderate horizontal speed (a few m/s) could give more suitable deployment conditions, but the result would depend strongly on the actual execution of the landing. Deployment after landing is therefore preferred.

5.2 Deployment after landing

Post-landing deployment is straightforward to analyse because of the absence of aerodynamic effects. For launch from ground level at elevation angle θ onto flat terrain, the range is $R = v_0^2 \sin(2\theta)/g$. At $\theta = 45^\circ$ the launch velocity required for a range R is

$$v_0 = \sqrt{Rg} = \sqrt{500 \text{ m} \times 0.113 \text{ m/s}^2} \approx 7.52 \text{ m/s} . \quad (6)$$

which is relatively benign. The corresponding time of flight is $t_f = \sqrt{2}v_0/g \approx 94$ s and the apex height is $R/4 = 125$ m.

On flat terrain the sub-payload impacts with the same speed as it was launched. This holds only for an impact point at the launch elevation; on sloped or uneven terrain impact speed and range change. The kinetic energy required to give a 0.5 kg sub-payload this speed is

$$E = \frac{1}{2}mv_0^2 = \frac{1}{2}mgR \approx 14.1 \text{ J} . \quad (7)$$

Characteristic launch speeds and energies for several payload mass and range combinations are given in Table 3 and Table 4.

Table 3: Launch velocity at 45° elevation versus range ($g = 0.113 \text{ m/s}^2$).

Range (m)	100	200	300	400	500	600	700	800	900	1000
v_0 (m/s)	3.36	4.75	5.82	6.72	7.52	8.23	8.89	9.51	10.08	10.63

Table 4: Sub-payload kinetic energy at release (J) versus mass and range, 45° elevation. Since $E = mgR/2$, the table is linear in both mass and range.

$m \setminus \text{range}$	100 m	200 m	300 m	400 m	500 m	600 m	700 m	800 m	900 m	1000 m
0.2 kg	1.1	2.3	3.4	4.5	5.7	6.8	7.9	9.0	10.2	11.3
0.3 kg	1.7	3.4	5.1	6.8	8.5	10.2	11.9	13.6	15.3	16.9
0.4 kg	2.3	4.5	6.8	9.0	11.3	13.6	15.8	18.1	20.3	22.6
0.5 kg	2.8	5.7	8.5	11.3	14.1	16.9	19.8	22.6	25.4	28.2
0.6 kg	3.4	6.8	10.2	13.6	16.9	20.3	23.7	27.1	30.5	33.9
0.7 kg	4.0	7.9	11.9	15.8	19.8	23.7	27.7	31.6	35.6	39.6

Because $R \propto v_0^2 \sin 2\theta$, the range is first-order insensitive to the elevation angle at 45° but scales linearly with the delivered energy:

$$\frac{\Delta R}{R} = \frac{\Delta E}{E} = 2 \frac{\Delta v_0}{v_0}, \quad \frac{R(\theta)}{R(45^\circ)} = \sin(2\theta). \quad (8)$$

The acceptable landing area of ± 100 m at 500 m ($\pm 20\%$, Section 7.4) therefore allows about $\pm 20\%$ in delivered energy, or $\pm 10\%$ in launch speed. Elevation errors from lander tilt are given in Table 5. Since only 250 m separation is required, sizing for a 500 m nominal throw covers lander tilts up to about 30° and an energy shortfall of up to 50%. Recoil on the lander is $mv_0 \approx 3.8 \text{ N s}$, which gives the 600 kg lander a velocity change of about 6 mm/s and is negligible unless the thrust line is far from the lander's centre of mass on a soft footing.

Table 5: Range reduction due to elevation error at 45° nominal launch angle.

Elevation error (lander tilt)	0°	±5°	±10°	±15°	±20°
Range factor $\sin 2\theta$	1.000	0.985	0.940	0.866	0.766
Range for 500 m nominal (m)	500	492	470	433	383

6 Discussion and selection of the design baseline

Deployment during descent poses significant engineering challenges because of the impact conditions, and it also couples strongly to operational aspects of the mission. The remainder of this note therefore focuses on a first design iteration of the post-landing deployment concept, in particular the ejection system.

Table 6: Summary of the concept trade.

Criterion	Deployment during descent	Deployment after landing
Reach for 250–500 m	Requires release at ≈ 50 m/s	7.5 m/s launch, 14 J per 0.5 kg
Impact speed	$\approx$ release speed (≈ 50 m/s)	$\approx$ launch speed (5–8 m/s)
Node azimuth distribution	Along approach axis only	Set by pedestal azimuths
Coupling to descent operations	Strong (timing, plume, collision)	None
Stored energy on lander	None	≈ 14 –20 J per node
Dependence on lander landing attitude	None	Range and azimuth
Assessment	Rejected for baseline	Selected baseline

7 Design of the post-landing deployment system

This chapter documents a first design iteration of a potential post-landing deployment system (PLDS).

7.1 Means of propulsion

Given the speeds and energies identified above, several means of propulsion are of potential interest to push the sub-payload away from the lander:

- spring;
- resistojet;
- cold gas;
- surplus pressurant from the main lander propulsion system (if applicable);
- electromagnetic launcher (“rail gun”).

Mass is of utmost importance for the Enceladus lander, so the energy density of the propulsion methods is an important trade quantity. The spring has the lowest storage energy density, about 200–400 J/kg for steel and up to about 500 J/kg for titanium alloys (wire only, at allowable shear stress), which is low compared with storing the energy in a lithium battery for electromagnetic or resistojet propulsion (≈ 500 kJ/kg), with cold gas in between (≈ 80 kJ/kg including tank). None of these values includes the hardware mass of the full system or the conversion efficiency, and for these aspects the picture is reversed. The spring only needs a latch as an additional element and can transfer a large fraction of its stored energy to the sub-payload (up to about 90%). For the other methods the efficiency is much lower, e.g. only around 1% for a cold-gas thruster.

From a preliminary back-of-the-envelope analysis it is concluded that a cold-gas system could, with high likelihood, be the lightest solution, followed closely by a resistojet.

The resistojet has particular advantages: its thrust can be regulated according to in-situ needs, e.g. to compensate for terrain or landing attitude. However, sublimating or organic propellants such as naphthalene would contaminate the landing site and can be almost ruled out for that reason alone, with purified water as the possible exception.

It should be noted, however, that the main lander’s descent engines will already contaminate the landing site with their own exhaust products, and in much larger quantities than any sub-payload propulsion system. Whether a small additional contribution from a “contaminating” propulsion solution is actually decisive therefore needs a closer look, together with the lander’s contamination budget, before a final decision is made.

A cold-gas thruster could in principle use noble gases (e.g. neon), which reduces contamination issues. Its main downside is the reliability of valves and seals over a long mission lifetime in a challenging environment. Electromagnetic systems are believed to be considerably heavier and add complexity.

Because of its inherent simplicity, the first design iteration uses a spring system. It also provides a reference for a later evaluation of how much heavier it really is than a cold-gas or resistojet solution.

7.2 Mechanical and functional concept

Because of the strict mass limits, the concept must be as simple as possible and omit everything that is not strictly needed. From a functional perspective the system consists of:

- the sub-payload module (0.5U, 370 g);
- spring(s);
- guidance features;
- a hold-down and release mechanism (HDRM);
- a pedestal, i.e. the mechanical interface to the lander that also sets the launch attitude.

Hardware possibly needed to manage the impact at landing is not considered at this stage.

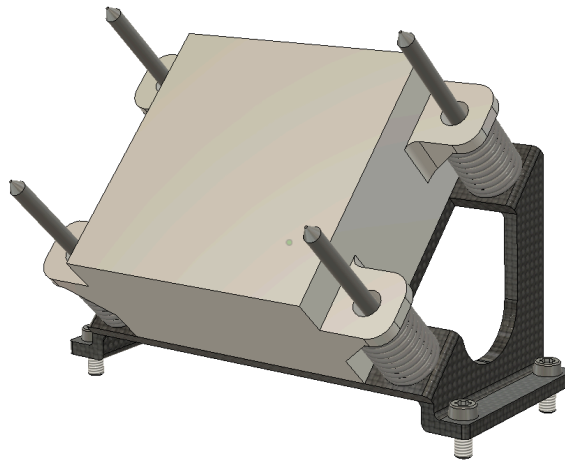


Figure 2: GPR sub-payload deployment concept 1: 0.5U node on a 45° CFRP pedestal with four compression springs and guide pins.

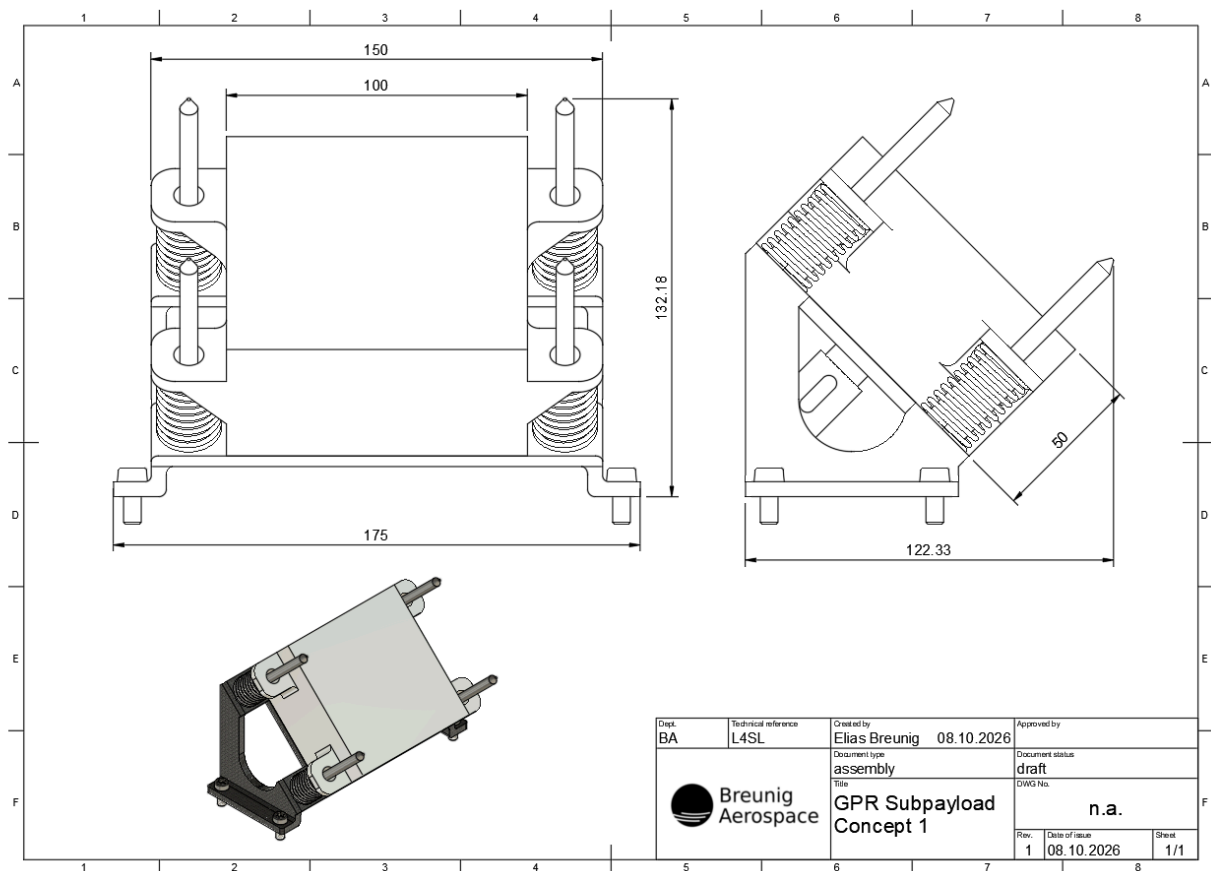


Figure 3: Assembly drawing of GPR sub-payload concept 1 (L4SL, rev. 1, 08.10.2026). Envelope 175 × 132 × 122 mm.

7.3 Springs

Distributing the spring function over four identical springs around the circumference of the 0.5U sub-payload box keeps the overall size increase modest. With a payload box height of 50 mm, the goal is to size the springs for a similar compressed length. Two variants are considered, a 1:2 and a 1:3 expansion ratio, i.e. a stroke x of 50 mm or 100 mm. For a linear spring released from full compression to its free length, the stored energy is $E = \frac{1}{2}kx^2$ and the ideal peak force is

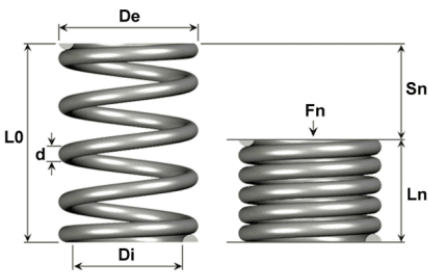
$$F = kx = \frac{2E}{x}. \quad (9)$$

With $E = 14.1$ J and $x = 0.05$ m or 0.10 m this gives $F = 564$ N or 282 N for all springs together. From engineering judgement the lower value is more attractive, since it would still allow the unit to be mounted and preloaded by hand (with a simple fixture), which is attractive at least for early prototyping. The corresponding peak acceleration of the 1:3 variant is

$$a = \frac{F}{m} = \frac{282 \text{ N}}{0.5 \text{ kg}} \approx 564 \text{ m/s}^2 \approx 57g_E, \quad (10)$$

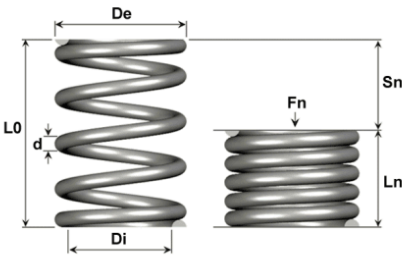
which is manageable for a compact payload.

A commercial off-the-shelf steel spring that delivers the required performance is the Sodemann Federn model 12570 (Figure 4). The datasheet gives a stored energy of about 3.5 J per spring, i.e. about 14 J for four springs, which is an almost ideal match to the requirement, although real-world losses would require about 20% margin. This could be achieved with the slightly larger model 22820 (Figure 5).



Material	Pianodraht
d - Drahtdicke (mm)	1.60
De - Außen durch-messer (mm)	21.60
Di - Innen durch-messer (mm)	18.40
L0 - Unbelastete Länge (mm)	110.00
Ln - Max. belastete Länge (mm)	25.50
Sn - Max. Weg (mm)	84.50
Fn - Max. Belastung bei Ln (N)	84.83
R - Feder-konstante (N/mm)	0.99
Serie	A

Figure 4: Sodemann 12570, music wire.



Material	Edelstahl 302
d - Drahtdicke (mm)	2.00
De - Außen durch-messer (mm)	27.00
Di - Innen durch-messer (mm)	23.00
L0 - Unbelastete Länge (mm)	135.00
Ln - Max. belastete Länge (mm)	31.00
Sn - Max. Weg (mm)	104.00
Fn - Max. Belastung bei Ln (N)	106.30
R - Feder-konstante (N/mm)	1.02
Serie	A

Figure 5: Sodemann 22820, stainless steel 302.

As both the assumed mass of 0.5 kg and the 500 m range already carry margin, we decide here not to enter an oversizing spiral and proceed with the smaller spring type. It should be noted, however, that the 12570 is not ideal in terms of material: music wire (carbon steel) is not the preferred choice for a cryogenic environment and a long-term preloaded state. It is therefore considered a reference for sizing and prototyping only; ultimately, a custom titanium spring of equivalent performance is the best choice, which also brings the lower mass already assumed in the mass budget (Table 7).

7.4 Guidance

The sub-payload probably needs guide rails or similar features. These are very likely needed for mounting and preloading onto the springs, but possibly much less for the deployment itself. The deployment does not need to be particularly accurate in terms of impact position: a landing area of up to 100 m radius at 500 m distance is potentially acceptable. It can therefore be postulated that accurate control of spring preload, spring rate and geometric tolerances is sufficient for a successful deployment, so that the guide rails or pins become MGSE. (See also Section 7.5 for another potential solution.)

7.5 Hold-down and release mechanism

A Frangibolt actuator is proposed to break the central attachment bolt that holds the sub-payload against the spring forces on the pedestal in a deterministic way. The Frangibolt actuator can potentially also act as a guide bushing, eliminating the need for the guide pins shown in the initial design concept. The Frangibolt housing is not shown in the CAD model but is included in the mass budget. It must be confirmed that a Frangibolt actuator can be made to tolerate the 75 K environment on Enceladus, as standard off-the-shelf models are only rated down to about 200 K. This is a major risk for the concept as proposed.

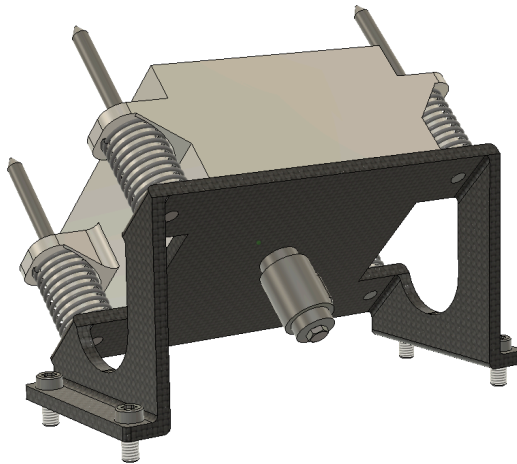


Figure 6: Rear view of concept 1 showing the Frangibolt FC3 actuator at the bottom centre of the pedestal.

Three further aspects need to be considered for the HDRM. First, a shape-memory actuator starting from 75 K needs considerably more heater energy and time to reach its transition temperature than at its qualification temperature, which must be included in the lander power budget. Second, the bolt preload must exceed the full spring force (about 280 N for the sizing in Section 7.3, more if the spring is oversized) plus the quasi-static launch and touchdown inertia loads of the node, to avoid gapping and rattling of the preloaded stack. Third, a low-mass alternative to the Frangibolt is a thermal-knife or burn-wire release with a low-creep tie (e.g. Vectran), as widely used on CubeSats; it weighs only a few grams, but must be checked for lifetime effects and issues under the prevailing environment.

7.6 Pedestal

The pedestal must be optimised for mass. A CFRP design with minimal metallic inserts is therefore likely to be the best solution. As shown, a further size reduction can be expected with a more evolved topology. If possible, a tight integration with other lander hardware and / or primary structure could yield further significant mass savings.

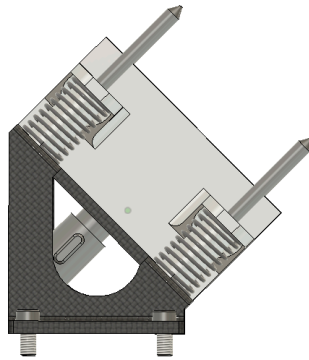


Figure 7: Side view of concept 1 showing the 45° CFRP pedestal.

7.7 Mass budget

The preliminary mass budget is given in Table 7.

Table 7: Preliminary mass budget of one GPR node with PLDS.

Pos.	Item	Mass (g)	Qty	Total (g)	Source / note
1	GPR payload	370	1	370	As specified by TU Dresden
2	Additional housing elements	40	1	40	CAD
3	Spring	7	4	28	Sodemann 12570, assuming titanium
4	Guide pins	18	4	72	CAD; lighter solution needed (potentially not needed)
5	HDRM bolt	5	1	5	CAD
6	Frangibolt actuator	40	1	40	Manufacturer datasheet
7	Frangibolt housing	20	1	20	Estimate, not yet designed
8	Pedestal incl. bolts	150	1	150	Extrapolated from CAD; believed to converge to ≤ 100 g
9	Actuator harness	10	1	10	Allocation TBC
Total per node				735	

The preliminary mass budget shows essentially a doubling of the mass of the bare sub-payload, which was expected from initial back-of-the-envelope estimates. From engineering judgement, a further reduction is possible, and a deployment-system mass of about 250 g seems realistic. With further mass optimisation of the sub-payload itself, a total of about 500 g for the integrated system is a realistic mid-term goal.

For the target of four nodes, a further mass saving can be expected from a shared structure, i.e. one pedestal carrying several nodes in different azimuths, which should be investigated in the next design iteration.

In general it is recommended to also seek out any mass-reduction opportunities within the actual sub-payload, as savings there will directly contribute to the overall mass reduction of the integrated system.